

LOW COST MOLNIYA SPACECRAFT FOR NAVIGATION/COMMUNICATIONS MISSIONS

J Radbone

Surrey Satellite Technology
Senate House
University of Surrey
Guildford
Surrey, GU2 5XH, U.K.

Tel: (0483) 571281 x 2279 Telex: 859331 Fax: (0483) 300803

Although the Molniya orbit has been widely used by the Soviet Union for many years in support of a variety of communications, science and navigation missions, its advantages have never really been exploited in the West. The Molniya orbit offers improved communications at high latitudes particularly for mobile communications, plays an important part in the NAVSAT navigation system constellation and offers scientists an opportunity to undertake scientific measurement over a variety of altitudes and hence environmental conditions with a single spacecraft.

Surrey Satellite Technology have recently completed a study to assess the feasibility, and then to propose a preliminary design for a low cost Molniya spacecraft which supports a navigation payload using a C/L band transponder of 60kg and 150W with growth potential for future missions. This paper presents the conclusions of that study and includes brief discussions on the launch and spacecraft options and the proposed solutions for a cost effective system.

INTRODUCTION

To implement continuous global navigation or communications systems a multiple satellite constellation of as many as 18 spacecraft are required. For example, to calculate 3 dimensional positions using space navigation systems such as GPS/NAVSTAR, which operate on the time of arrival principle (TOA), signals from four satellites must be received

simultaneously. Hence a constellation of typically 18 satellites is necessary to maintain the required global coverage in either circular, or geostationary and highly elliptical orbits.

Since so many satellites are involved in such a system, any decrease in the cost of each satellite will greatly decrease the overall cost of the system.

This paper describes the results of a study carried out to investigate the design of a low cost Molniya spacecraft which forms a segment of the European navigation system NAVSAT, and which also has applications for provision of communications at high latitudes. The study was specifically to determine whether a Molniya satellite could be built using the cost-effective engineering methods pioneered at the University of Surrey originally proved in Low Earth Orbit through the UoSAT programme.

UoSAT LOW COST DESIGN PHILOSOPHIES

The following low cost design philosophies were the basis for the design of the UoSAT-1 and 2 spacecraft, designed built and operated in orbit by the University of Surrey.

Keep It Simple - Design to meet the task, not just 'gold plate'.

Complete engineering understanding of the requirements and realistic safety margins should form the basis for the selection of components and design techniques, the expense of high quality/ratings approaches is not always justified.

Use Standard, Proven Designs for Essential Modules

The UoSAT spacecraft are based on a hierarchical layered approach where the essential housekeeping modules use standard, proven designs and hardware where possible, and levels of sophistication are added as increasingly higher levels using newer, unproven technology as a 'hot' redundant paths.

Redundancy via Flexible Design and Not Just Duplication

Provision of alternative data routes using different technologies gives protection against not only random failures but also possible design anomalies.

Simple Interfaces

Minimising interface documentation and assembly time by using simple interfaces will help keep the costs to a minimum.

Use Established, Volume Production Components

The UoSAT experience has been that in many cases, the reliability of commercial, volume production, low cost components is comparable with the High Reliability, and high cost, components produced in small quantities for the space industry, and with basic screening may be used successfully in many space missions.

BASELINE SPACECRAFT PAYLOAD SPECIFICATION

The satellite was designed to carry a payload with the following characteristics:

Payload:	C/L Band Transponder operational at altitudes above 10635km
Mass:	60kg (including antennas)
Power:	150W average
Pointing:	+/- 1 degree when operational
Lifetime:	> 3 years
Antennas:	L-band - 61 element, 2m diameter array C-band - horn, 0.07m diameter x 0.17m

LAUNCH OPTIONS

Whilst considering a low cost spacecraft it is important not to forget the launch costs and aim to maintain them in keeping with the satellite cost.

The Molniya orbit has traditionally been the USSR speciality and has rarely been attempted in the West.

To achieve the correct Molniya orbit several orbit parameters must be achieved in addition to the correct apogee and perigee. Correct inclination (i.e. 63.45) to provide the constant ground track and positioning of the apogee is over the required sub-satellite point i.e. that the Right Ascension of Ascending Node (RAAN) is correctly fixed in inertial space, must also be ensured. In addition the apogee must be placed such that it is at the maximum height over either the northern or southern hemisphere i.e. the argument of perigee must be set to either 90 deg (southern) or 270 deg (northern).

Launchers such as Conestoga and AMROC appear to offer the opportunity for lower cost launches more in keeping with the likely cost of the spacecraft, approximately £5-6M. They would carry a single spacecraft per launch and so minimise mass penalties for adapters etc and more complex launcher manoeuvres. However, at this stage, they are not proven, reliable technology and so for the purposes of the study Delta and Ariane were considered in more detail.

Since Ariane IV does not have a re-ignitable second stage, the satellite would have to carry an Apogee Boost Motor to reach the final orbit. The disadvantages of carrying an ABM are many - extra cost, large volume requirement, heavier structure to survive the additional g forces and carry the extra weight, and dead weight carried in the final orbit. Consequently the Delta launcher which does have a re-ignitable second stage (with at least 6 re-fires) and is able to place a spacecraft direct into the required orbit and was chosen as the preferred option.

The Delta 7920 (liquid + 9 solid boosters first stage, liquid second stage and no third stage) can put 640kg into the final orbit and the 7925 (as the 7920 plus PAM-D third stage) can place 1156kg into the Molniya orbit. Based on a mass budget where the total spacecraft mass (wet) was approximately 270kg, a dual launch on the 7920 is proposed as a feasible and most cost effective option.

Delta Launch Scenario

Launch would be from the West Test Range (WTR) which usually offers angles between 80 deg and 145 deg. To achieve the required 63.45 deg inclination, the vehicle would launch at 185 deg followed by a dog leg manoeuvre to the correct inclination.

Considering a dual launch scenario some separation in mean anomaly may be required. Additional capacity of the launcher could provide up to 24 degrees separation in mean anomaly around the orbit. Additional separation could be carried out using the on-board reaction control system (RCS) with an acceptably low overhead in fuel mass (the exact amount being dependant on the time allowed for the manoeuvre). e.g separation of 60 degrees in 24 hours requires delta V of approx 50 m/s.

SPACECRAFT CONFIGURATION OPTIONS

Four spacecraft configuration options were considered to assess which attitude control scenario would meet the requirements in the most cost effective manner. These options were:

- Option A 3 axis stabilised with all antennas on one face maintained Earth pointing. Solar cells on deployable panels steered for maximum illumination.
- Option B Like option A except with body mounted arrays instead of steered wings.
- Option C Dual spin - body spinning with antennas on despun platform. There are two options within option C with the spin axis either normal to the orbit plane (Hughes type) or spin axis in the orbit plane.

Option D Gravity gradient stabilisation, Earth pointing. (UoSAT type technology)

Option E Spinning spacecraft whose pointing is fixed inertially in space. The only time it would be Earth pointing is at apogee.

Option E was rejected for this particular payload since the communications link budget was unacceptable at the extremes of the operational phase. Only with another 5dB communications margin could this become an acceptable option. This option was, however, perhaps the most attractive in terms of low costs and indeed this technique has already been successfully used by the Phase 3 Radio Amateur spacecraft and has potential applications for other low cost Molniya missions.

Option D can also be rejected as gravity gradient techniques are not stable for orbits with ellipticity above 0.355. This means the current UoSAT attitude control systems of passive gravity gradient and active magnetorquing are not appropriate for this orbit however, as will be shown in the ADCS Sub-System section later in this paper, magnetorquers still have a role to play in the proposed system design.

Of the other three possibilities Option B was selected as the most cost effective for the following reasons:

Simplest geometry and construction will minimise costs
No mechanisms will improve reliability and reduce costs
Body mounted arrays does not inhibit dual launch in a Delta
Small mass disadvantage to achieve simplicity is not inhibitive
Simple attitude determination and control sensors and scenario
Simple groundstation operations

SPACECRAFT BUS DESIGN

The results of the study trade-offs are basically summarised in the final proposed system block diagram shown in Fig 1. The rest of the paper looks briefly at the trade-offs and proposals in each of the sub-system areas.

Communications Sub-System Design

The payload is a navigation communications C/L band transponder. In normal operations telemetry and telecommand will also use the payload C band antenna for up and down link communications. The satellite control during the transfer orbits and initial stabilisation such that it can also be carried out using the main C band antenna system should this be necessary or preferable.

However, if the transfer stage does not go to plan or there is some anomalous conditions it is proposed to fly an emergency C-band antenna which has a 'pancake' shaped response. e.g. a slotted cylinder arrangement deployed after launch.

Data Handling

In normal operations, the data handling sub-system will be controlled by the On-Board Computer (OBC) and will be responsible for

- * decoding telecommands, verifying them (checking for undesirable command strings, user passwords etc) and then implementing them by setting the appropriate commands latches.
- * formatting the telemetry packets in either standard format (all parameters updated each frame) or providing 'dwell' telemetry where selected channels are downloaded more frequently. Appropriate coding will be included in the Telemetry frame to provide immunity from transmission errors.
- * executing the attitude determination and control algorithms.
- * self-checking of spacecraft general health and performance.

A high level of autonomous operations on board the spacecraft will enable the groundstation costs to be minimised and sophisticated on-board control.

Should there be a failure of the OBC or associated devices this would not mean the end of the mission. There will be the 'minimum' hardwired telemetry and telecommand system that would enable the command latches to be driven directly from the hardwired telecommand system such that all spacecraft operations could be continued under manual control.

This level of autonomy and failure resilience has already been proven on the UoSAT spacecraft and has provided high levels of reliability and performance for missions exceeding 5 years.

Some care must be taken in choice of semi-conductor devices for use in this orbit where the radiation damage will be greater than both low Earth and geostationary orbits since the satellite spends more time in the Van Allen Belts. Based on ESA software predictions, the worst case overall fluxes (including conversion of proton damage to equivalent electron damage) that might be expected are 9.41×10^{14} electrons/sq.cm over 3 years, and with a 4mm 'Al Slab' protection the total dose/year is 2.5krad, i.e 7.5 krad over 3 years.

Use of suitable technologies and precautions (eg EDAC) can protect against this level of radiation adequately for lifetimes exceeding the specified 3 years.

Telemetry and Telecommand Sub-System

The first estimate of the telemetry and telecommand data requirements is by no means fixed. It represents the order of magnitude of the data handling requirement which can be served quite adequately by a downlink data rate of 1200 bps with frame updates at typically 2s intervals if required. sync bits/byte) at 1200 bps the max update rate is 1.22 seconds (giving approx 100 analogue channels and 80 digital status points). If a very fast update rate was required one could use a dwell facility where only one (or a few) specified telemetry channels are transmitted. This type of system has already been implemented on both UoSAT spacecraft and the dwell facility gives not only a means for fast analysis of housekeeping sub-system performance but also enables the telemetry system to act as a channel for higher data rate experimental payloads.

Power Sub-System

The payload requires an average of 150W in its operational phase (i.e above altitudes of 10635km), which is comprised of a continuous 85W element and a pulsed element of 1120W for 140ms every 2.4 seconds. This 1120W provides 280W RF power (25 per cent efficiency). In the non-operational phase around perigee, the payload will require 10W stand-by and the other 140W will be available to drive the magnetorquers as part of the attitude control system.

The rest of the systems require 30W average which results in the solar panels having to provide 180W orbit average.

Based on solar illumination and power system efficiency analysis the spacecraft array area and battery capacity was calculated trading off silicon and gallium arsenide, and Nickel Cadmium and Nickel Hydrogen technologies.

Silicon was selected for the solar arrays since despite the thermal losses and lower efficiency the overall size still allowed dual spacecraft to fit comfortably within the selected launcher. The additional cost of gallium arsenide, 8-10 times that of silicon, would only be recommended if the launcher envelope was restricted or the thermal environment much more stringent.

Nickel cadmium batteries were also selected as the most cost effective option with current technology despite the mass disadvantage compared to Nickel Hydrogen cells. Nickel Hydrogen would only be recommended if the mass became critical since they are typically 8-10 times more expensive.

The resultant system therefore comprises a spacecraft of approximately 1.8m height and 2m diameter. The solar arrays are arranged on 12 sided cylinder for ease of manufacture compared to a curved surface.

A bus voltage of 28V is chosen as being high enough that the voltage drop due to the pulsing of the payload ($<4V$) will not effect the operation of the spacecraft bus DC-DC converters and low enough that the component technology is readily available.

The resulting battery mass for NiCd cells is approx 22kg which can be comfortably accommodated within the mass budget and hence NiCd cells offer the most cost effective technology.

To maximise the power output from the solar arrays, based on the sun aspect angle and the array temperature, ideally each panel should have its own battery charge regulator. However, adjacent panels are considered to be under almost identical conditions and hence six BCR's are proposed each with temperature dependant peak power tracking for maximum power output.

The power for the payload will be taken directly from the battery to provide isolation of the stabilised voltage rails from the drop in bus voltage due to the payload pulsing requirements. The internal resistance of the battery is approx 2-4 mohm/cell which will result in a drop in bus voltage of 2-4V which can be accommodated in the voltage converters.

UoSAT has used silicon arrays and nickel cadmium batteries on both its spacecraft together with automatic temperature tracking BCR's. The nickel cadmium batteries used on UoSAT-2 were an example of using commercial grade, high volume components with basic screening and matching as a low cost alternative to the space qualified batteries used on UoSAT-1. No significant difference in performance has been observed over UoSAT-2's four and a half year lifetime to date.

Attitude Determination and Control Sub-System

The basic requirement is to maintain Earth pointing. The satellite achieves this by dual-spin stabilisation - an internal momentum wheel whose angular momentum is aligned on the pitch (Z) axis of the satellite and points nominally in the direction of the orbit normal. Controlled speed changes maintain Earth pointing and a nominal pitch angle of zero.

Attitude information is provided by a specially developed sensor which will handle the variable FOV presented by the Earth, during the active region of operation. This FOV varies between 16 degrees full angle at apogee to 44 degrees on the limit between active and inactive region. Because the Earth FOV widens even further during the inactive regions,

maximising at perigee and because - probably- the Earth sensor cannot make a measurement in this region a rate-integrating gyroscope (RIG) is required to back-up the pitch sensing during the inactive region.

The intention is to maintain Earth pointing during the inactive region so that there should be no re-acquisition problem on re-entering the active region.

The size of the momentum wheel required for initial stabilisation is estimated to be 30 Nms. This value was then verified as being sufficient to deal with the worst case disturbance torques due to firing of misaligned thrusters (assuming a 0.1 degree misalignment error).

Correction of any Z axis pointing error is carried out during the payload 'inactive' region. The roll error during the active phase will have been logged a computer program evaluates the current pointing error and then switches on magnetorquers to correct it. The pass through perigee is the only part of the orbit when the geomagnetic field is sufficiently strong (about 1/2 hour) to enable effective magnetorquing. This kind of technique has been proved by OSCAR-10 in a similar orbit.

Magnetorquing coils will also be employed to dump the momentum wheel angular momentum. Since volume is not restricted in this spacecraft, the advantage lies with aluminium 'air' spaced coils. Typically a magnetic moment of 1000 A.m² can be achieved, with a few kilograms of wire, and a power consumption maximally available of 140W. This specification enables either the entire momentum of the satellite to be dumped or the satellite slewed through 90 degrees if need be during just one perigee pass.

Although UoSAT have not as yet had any direct experience in using momentum wheels they have extensive experience in the use of magnetorquing and use of sophisticated control algorithms carried out by on-board computers to achieve simple and low cost operations.

ADCS Acquisition Sequence

The Delta launcher will deploy both spacecraft (slightly separated) with their spin axes normal to the orbit plane in a near 12 hour final orbit. The momentum wheel is then spun up causing the spacecraft to transfer momentum so that the spin axis is in the orbit plane but with a slow tumble. The wheel speed is then controlled to enable locking to the Earth using the Earth sensor to control the spacecraft pitch.

After the attitude has been stabilised then the hydrazine system can be used to separate the satellites the required amount in mean anomaly by firing along the roll (Y) axis. If the original orbits period was 11.8 hours, the time between firings for maximum separation of 2 hours would be 10 orbits, i.e 5 days.

Reaction Control Sub-System

When the spacecraft is used as part of a larger constellation of satellites to provide global coverage, it is important that the satellites do not drift relative to each other. Since the drift is dependant on specific orbit parameters (eg RAAN) each satellite will drift differently and would quickly cause loss of global coverage unless some orbit maintenance is carried out.

There is still much discussion as to the exact orbital perturbations and resulting changes in the Molniya orbit, however it appears that the station keeping requirements will be comparable with the N-S station keeping for geostationary orbits, i.e. a velocity increment of approximately 50 m/s per year. The RCS system below has therefore been designed to provide a delta V of 150 m/s minimum (3 year lifetime).

The RCS system is also required to enable the satellites in the dual launch to be separated in mean anomaly around the orbit. The amount of fuel required for this operation is directly related to the time allowed for the overall manoeuvre. The longer the period over which the delay can be built up, the closer to the final orbit the spacecraft can be during this 'drifting apart', and hence the smaller the fuel required to return them to the final orbit. For the sake of this system design it has been assumed that the separation can be achieved in less than a 24 hour period. The resultant fuel required is approx 50 m/s.

Cold gas, mono-propellant and bi-propellant were considered but it was a fairly simple trade-off which resulted in the selection of mono-propellant hydrazine as the proposed solution which would result in a reasonable mass budget using proven technology for the lowest cost.

Use of surface tension tanks which promise to overcome the traditional problems of failure due to bladder decomposition, despite being more expensive single items enable a single tank system to be proposed which will keep the overall sub-system costs low whilst maintaining sufficient reliability.

Seven 2 Newton thrusters are proposed to fire in all axes to correct for drifts in argument of perigee and RAAN. Larger 20 Newton thrusters are proposed for the initial separation in mean anomaly to reduce the required burn time to the order of 10 minutes. These thrusters could also be used as back ups for the station keeping thrusters.

The currently available small surface tension tanks are typically 38 dm³ tank with an internal diameter of approx 400mm. This size of tank has been designed into the structure and enables extra fuel to be carried to extend the lifetime beyond the 3 years specified.

Thermal Sub-System

The thermal analysis showed that low cost techniques of controlling surface finishes together with small line and patch heaters and Rockwool type insulation for the hydrazine system would enable sufficient thermal control to maintain the solar array efficiency and dissipate the payloads heat output. The ends of the spacecraft would use black and white paint to provide an absorptivity of 0.45 and an emissivity of 0.88.

The crux of this analysis is the 'evening out' of the temperatures which is clearly dependent on the construction and materials of the array panels and structure. This analysis has aimed to show that thermal control based on low cost, basically passive techniques is a realistic proposal however this was only a preliminary analysis and would need refining as the structure became better defined.

Structure

The satellite structure is required to do the following:

- a) support the lower satellite on the launcher
- b) support the upper satellite during launch
- c) support all the spacecraft components
- d) be light, easy to make, easy to install components, robust, low cost.

The major stresses on the structure will occur during launch or ground handling. The launch criteria are:

- i) axial acceleration of 6g (x1.25) plus vibration
- ii) lateral acceleration of 2.5g (x 1.25) plus 'shaking'
- iii) vibration and shaking through defined frequency ranges
- iv) qualification tests to 1.25 design loads plus no failures below 1.4 of design load
- v) ground handling will require lifting points etc, 3g loading should suffice
- vi) shock loading when explosive bolts are fired, local mounting stresses for individual components will have to be checked.

The two options analysed were

1. a 3 strutted space frame with triangular cross bracing on the top and bottom.
2. a cylinder 'thrust column' structure with bracing on the base to transfer the load to the PAF.

The optimum structure proposed was a hybrid of the two options where the main stress was taken by the 3 upright struts which were inclined to enable stress transfer into the Payload Attach Fitting (PAF). The solar arrays and top and bottom plates are only required to carry their local loads. The mass of the structure is estimated at 39 kg and enables a dual launch situation. The necessary sub-system modules fill a small percentage of the available volume allowing for future growth of fuel tanks or additional payload volume.

Structural material based on aluminium alloys (as used for UoSAT spacecraft) are proposed rather than composite structures for a low cost and easy to manufacture structure where mass is not critical.

Groundstation and Operations

It is often the case that the groundstation operations over the lifetime of a spacecraft is equivalent to the costs associated with actually getting the spacecraft launched. As a result it is as important to consider low cost groundstation operations as it is to consider low cost satellites.

The proposed satellite design has been proposed not only as the most cost effective satellite option but as providing the most cost effective overall performance. This is achieved by providing the means to make the satellite and groundstation as autonomous as possible, by simplifying all the data communications in terms of data formats and speeds and by making the satellite attitude control as simple as possible.

CONCLUSIONS

At the end of the study it is possible to conclude that, yes, many of the UoSAT philosophies, demonstrated by the UoSAT spacecraft, can be applied to the design of high elliptic orbit spacecraft for satellite navigation systems, and indeed many other applications.

The proposed 3 axis stabilised design with body mounted solar arrays is highly flexible running under the control of an intelligent on board computer whose software may be uploaded and revised throughout the mission. It has growth potential with considerable free volume and 96kg mass margin which could be used to carry more modules or increase the fuel quota to prolong the lifetime. The baseline design could be used not only for navigation payloads but also for scientific satellites and communications spacecraft with modest power requirements.

The spacecraft has been designed with low cost groundstation operations in mind using simple data formats, autonomous operations and simple attitude control techniques.

Finally it should be mentioned that this has been a preliminary study to assess the feasibility of a design and that further design stages are necessary to confirm the results and assess how low cost manufacturing and 'industrialisation' that will translate the low cost satellite to a truly low cost programme.

ACKNOWLEDGEMENTS

The work included in this paper was undertaken by Surrey Satellite Technology sponsored by the European Space Agency. Surrey Satellite Technology (SST) was formed to provide a professional interface between the University of Surrey Spacecraft Engineering Unit (UoSAT) and industry to enable efficient technology transfer and commercial development of on-going research and to undertake industrial contracts. SST would like to thank all the members of the UoSAT Unit and GTS who were contributed to this study.

REFERENCES

- (1) NAVSAT GEO + HEO Configuration - Final Report, Vols 1 & 2
ESA Contract No. 6354/85/F/RD(SC)
Italspazio, May 1986
- (2) T-SAT - Report on the Technology Satellite Design Study Vol 1
Science and Engineering Research Council, October 1986
- (3) "Optimum Yaw Matrix for Satellite with a Nadir Pointing
Payload"
Kalweit, C.C., J.Guidance and Control, Vol 6 no. 1, Jan/Feb
1983
- (4) AMSAT/OSCAR-10, launched June 1983, HEO (Apogee 35 500km,
Perigee 3 955km, Incl. 26 deg)
For brief technical overview see "Satellite Experimenter's
Handbook", M.R.Davidoff ARRL, 1984
- (5) AMSAT/OSCAR-13

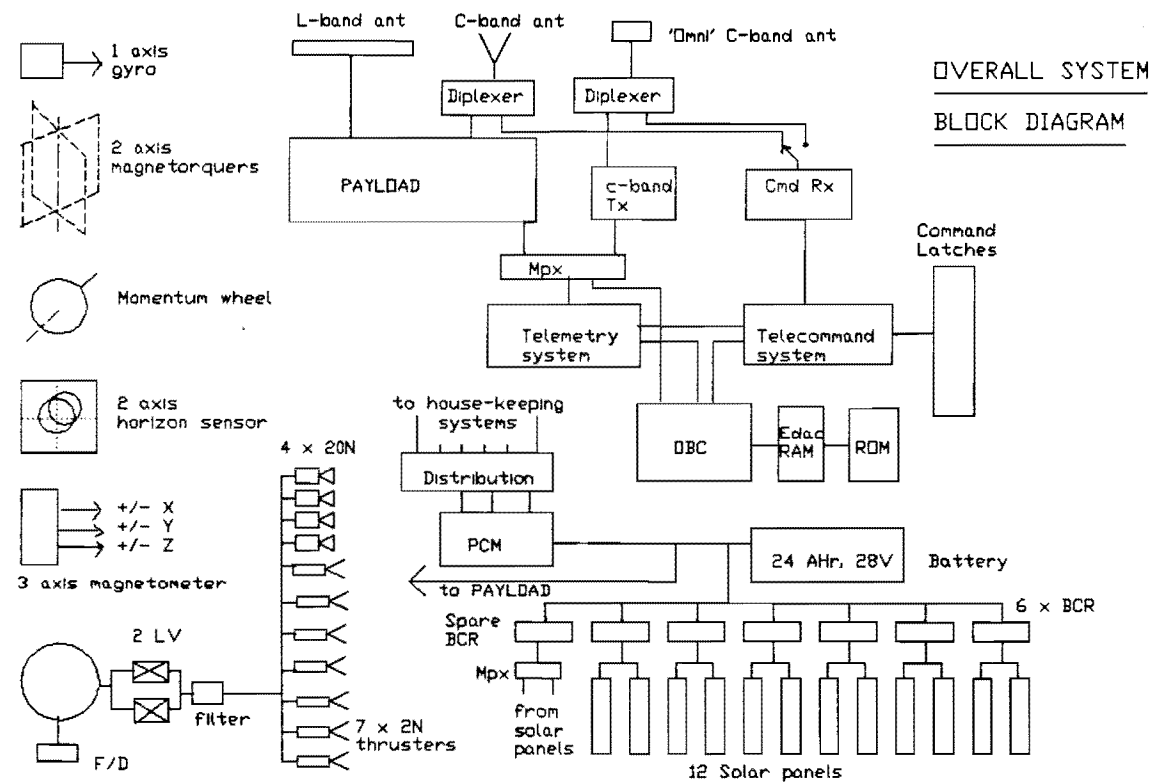


FIGURE 1 LOW COST MOLNIYA SATELLITE FOR NAVIGATION BLOCK DIAGRAM